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RESEARCH MEMORANDUM

MEASUREMENTS OF THE CHORDWISE PRESSURE DISTRIBUTIONS OVER THE WING OF THE XS-1 RESEARCH AIRPLANE IN FLIGHT

By De E. Beeler, Milton D. McLaughlin, and Dorothy C. Clift

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NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS
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SUMMARY

Measurements of the chordwise pressure distribution over the 8-percent-thick wing of the XS-1 research airplane have been made at a section near the midspan of the left wing. Data presented are for a Mach number range of 0.75 to 1.25 at a normal-force coefficient of about 0.33 and for normal-force coefficients up to 0.93 at a Mach number of approximately 1.16. The results show that there is a rearward shift of section center of load with an increase in Mach number due to the rearward movement of shock with a corresponding extension of the region of supersonic flow. The load center moves from about 25 to 51 percent of the chord as the Mach number is increased from 0.75 to 1.25. During the rearward movement of load from the forward to rearward limit position, there is a rapid and large shift of the center of load within these limits for a Mach number range of 0.82 to 0.88. It is expected that large changes in trim, with corresponding large changes in load factor at low altitude, may occur within this Mach number range.

The distribution of load over the section up to a Mach number of 0.95 is somewhat irregular and the irregularities are more pronounced on the upper surface. At Mach numbers from 0.95 to 1.25 the shapes of the distributions of load over both surfaces are somewhat similar. At high normal-force coefficients, however, the upper-surface pressure distribution approaches a rectangular distribution. A section normal-force coefficient of 0.93 was reached at a free-stream Mach number of about 1.16, with no indication that maximum lift had been reached. A maximum local Mach number of about 2 was obtained at this point.

There is a gradual transition of the total load distribution from one having peak loads near the leading edge at subsonic speeds to one approximating a rectangular shape at supersonic speeds. Distributions for Mach numbers above 0.95 are somewhat similar in shape.

Large changes in the pressure distribution over the rear 15 percent of the chord within a Mach number range of 0.80 and 1.0 would indicate large changes in hinge moments for an aileron at this location.

INTRODUCTION

The NACA and the Air Force are engaged cooperatively in an accelerated flight-testing program to establish the operational limits of the XS-1 research airplane. The airplane is flown by Captain Charles Yeager of the Flight Test Division at Wright Field and the flight data are obtained from NACA instrumentation and are evaluated by NACA personnel.

During the early stages of the test program of exploring the transonic speed region, sufficient instrumentation was installed to determine only the general behavior of the airplane and some loads on the wing and tail. As the flights progressed to higher speeds and repeated flights were successfully completed above the speed of sound (references 1 and 2), changes in instrumentation were made to obtain additional aerodynamic data. Measurements of chordwise pressure distribution over the midsemispan wing of the airplane have been obtained at Mach numbers greater than 1.0. Results of these measurements are reported herein.

SYMBOLS

M free-stream Mach number $\left(\frac{\text{True velocity}}{\text{Sonic velocity}}\right)$

M_L local Mach number

n normal load factor

W airplane weight, pounds

S wing area, square feet

C_{N_A} airplane normal-force coefficient $\left(\frac{nW}{qS}\right)$

b wing span, feet

c local wing chord parallel to plane of symmetry

x	chordwise location from leading edge, feet
q	free-stream dynamic pressure, pounds per square foot
p_o	free-stream static pressure, pounds per square foot
p_t	free-stream total pressure, pounds per square foot
p	local static pressure, pounds per square foot
P	pressure coefficient $\left(\frac{p - p_o}{q}\right)$
P_{cr}	pressure coefficient for sonic velocity
P_U	pressure coefficient on upper surface
P_L	pressure coefficient on lower surface
P_R	resultant pressure coefficient $(P_U - P_L)$
c_n	section normal-force coefficient $\left(\int_0^{1.0} (P_L - P_U)\, d\left(\frac{x}{c}\right)\right)$
$c_{m_{c/4}}$	section pitching-moment coefficient, stall moment is positive $\left(\int_0^{1.0} (P_U - P_L)\left(\frac{x}{c} - 0.25\right) d\left(\frac{x}{c}\right)\right)$
c_h	section hinge-moment coefficient $\left(\frac{1}{0.0225}\int_{0.85}^{1.00} (P_U - P_L)\left(\frac{x}{c} - 0.85\right) d\left(\frac{x}{c}\right)\right)$

DESCRIPTION OF THE AIRPLANE AND WING SECTION

The XS-1 research airplane assigned to the accelerated flight program is shown in figure 1. A three-view drawing of the airplane showing the general over-all dimensions is given in figure 2.

The airplane has an 8-percent-thick wing and incorporates an NACA 65-108 airfoil section. The profile and ordinates of the section are presented in table I. The wing test section was located at approximately the midspan of the left wing inboard of the aileron station and included the landing flap (fig. 3). The area about the test section was polished during the tests but no attempt was made to obtain a perfectly smooth finish. Landing-flap junction and the upper-surface wing spoiler presented irregularities, as did the spar-skin attach points. Location of these irregularities are given in figure 3. The skin thickness at the test section is approximately $\frac{1}{4}$ inch thick on both the upper and lower surfaces.

INSTRUMENTATION

Standard NACA recording instruments were used to obtain airspeed, pressure altitude, and normal acceleration. Wing-surface pressures were measured by an NACA recording multiple pressure manometer. All records were synchronized by a common timer.

Free-stream static pressure was recorded from a Kollsman pitot-static fixed head located at the wing tip. The static vents were located approximately 0.96 of the local wing chord ahead of the wing. Dynamic pressure was measured from the wing-tip pitot-static head (fig. 3).

Wing-surface pressures were measured from flush-type orifices installed in the wing skin. The orifices were connected by $\frac{1}{8}$-inch inside diameter rubber tubing to the multiple manometer located in the fuselage behind the pilot's compartment. The average length of tubing from the orifice to the manometers was approximately 9 feet. The orifice locations are given in the table of figure 3.

Normal acceleration of the airplane was measured near the center of gravity of the airplane.

METHODS

The pressure cells of the multiple manometer were vented to the fuselage instrument compartment; thus, pressures of the wing surface were measured relative to the existing compartment static pressure. Static pressure at the pitot-static head was also measured relative to the compartment pressure. A continuous record of the geometric altitude during the test flight was made by the radar ground station which was synchronized with a recording pressure altimeter in the airplane. On completion of the test flight, a survey was made by the airplane through the test altitudes obtaining synchronized airplane pressure altitude and radar geometric altitude records. The survey was made at speeds where the error in measuring true static pressure by the airplane was known. By this method, pressures measured at the wing were corrected to values relative to the free-stream static pressure. Ground tests were made to determine any effects of lag that might be present in measuring the wing-surface pressures. These tests show that the effects of lag were negligible and have been neglected in these data.

TESTS

The data presented herein were obtained during an attempted four-rocket flight at 50,000 feet. Due to faulty operation of one rocket, the flight was made with three rockets on and the airplane dived from 47,000 feet to obtain higher speeds. Recovery was made from the dive at approximately 35,000 feet by use of the adjustable stabilizer and resulted in a normal acceleration of 5.7g at a Mach number of 1.13. Continuous records of pressure distribution were obtained during the climb from 30,000 feet to 47,000 feet at Mach numbers of 0.75 to 0.95, during the dive to 35,000 feet to a Mach number of 1.25, and in the recovery at 35,000 feet at a Mach number of approximately 1.16.

PRESENTATION OF DATA

Presented in figure 4 are pressure distributions for the upper and lower surface of the wing test section for a Mach number range of 0.748 to 1.088. These distributions have been selected for an airplane normal-force coefficient of about 0.33. Distributions for Mach numbers from 1.138 to 1.248 are given in figure 5 for an airplane normal-force coefficient of about 0.16. Included in the figures are values of the free-stream Mach number and pressure coefficients for sonic velocity for each distribution. It may be noted that there are two values of pressure coefficient given at about 10 percent of the chord on the upper surface between a Mach number of 0.75 and 0.85. These represent extreme values of the pressure coefficient during pressure fluctuations at the orifice. The pressures existing for the selected conditions are used for fairing the pressure distribution. Pressures at orifice stations forward and to the rear were steady.

The variation of the local pressure coefficients for the upper and lower surfaces with free-stream Mach number are presented in figures 6(a) to 6(d). Included in the figures are locations of the pressure stations in percent of local chord and, in some plots, local Mach number lines of 1.0 and 1.5 which are computed from equations for subsonic flow. It is estimated that the error in using the subsonic equation for local Mach number is less than 5 percent at the highest free-stream Mach number reached in these tests.

In order to show values of local Mach number regions existing over the airfoil section at a given free-stream Mach number, some pressure distributions for the upper and lower surfaces have been selected and are presented in figure 7 as p/p_t against percent of section chord. The scale of the local Mach number is given as the ordinate on the right of the figure. For these plots the local total head may be assumed to be equal to the free-stream total head. Errors introduced by this assumption are less than 1 percent for the maximum free-stream Mach number given in figure 7.

From the inspection of the pressure distributions of figures 4 and 5, which give the variation of local pressure coefficient with Mach number, and of the continuous records of individual pressures, the approximate location of the shock for both the upper and lower surfaces has been determined and is given in figure 8 plotted against free-stream Mach number. The termination of the supersonic flow over the forward portion of the airfoil by an abrupt pressure recovery has been taken primarily as indicative of shock.

Pressure distributions obtained during a dive pull-out are presented in figure 9 for various normal-force coefficients at a free-stream Mach number of approximately 1.16.

The total load distributions over the section have been derived from the faired distributions of individual surfaces and are presented in figure 10 for a Mach number range of 0.748 to 1.185 at an airplane normal-force coefficient of about 0.35. Included in the figure are values of the free-stream Mach number and the integrated section normal-force coefficient for each distribution.

A summary figure showing the wing section characteristics for the Mach number range presented is shown in figure 11. The section normal force, hinge moment, and center of pressure have been obtained from the mechanical integration of measured pressure distributions. The section pitching-moment coefficient has been derived from the center-of-pressure curve for two section normal-force coefficients. The section hinge moment as presented herein is the moment of the area of the rear 15 percent chord about the 85-percent-chord station. The hinge moments at Mach numbers less than 1.0 would be representative values for aileron neutral at a wing section which included the aileron. Hinge moments for Mach numbers greater than 1.0 may not be representative due to wing-tip losses.

DISCUSSION

The free-stream Mach number at which local sonic velocity is first attained at any point on an airfoil is normally defined as the critical Mach number of the section, and the occurrence of the peak pressure coefficient attainable at any point on the airfoil is usually associated with the formation of a compression shock. The critical Mach number has not been established in these tests since, at the lowest Mach number reported $(M = 0.748)$, supersonic flow already exists over approximately 40 percent of the chord. (See figs. 4 and 7.) As shown by measurements of surface pressures over the airfoil (fig. 6) the pressure coefficients increase negatively with an increase in free-stream Mach number until some peak pressure coefficient is reached. As the free-stream Mach number is increased above the value where the peak pressure coefficient is reached, the local pressures increase positively tending to follow, in general, a constant local Mach number line.

Upper-Surface Pressure Distribution

An inspection of the pressure distribution presented in figures 4 and 5 for a given airplane normal-force coefficient shows that, as the Mach number is increased from 0.75 to 1.25, there is a gradual transition of load distribution over the section. The transition is from a compressible subsonic type of pressure distribution having peak values near the leading edge followed by a normal pressure recovery over the rear section to a supersonic type of pressure distribution having maximum negative pressures over the rear section. It may be seen that the transition results in a shift of the load center to the rear with increasing Mach number. As will be shown in the following discussion of selected Mach number ranges, the load center shift is a result of the shock movement toward the rear and the extension of the region of supersonic flow expansion over the airfoil.

<u>Mach numbers 0.75 to 0.82</u>.- As the Mach number is increased from 0.75 to 0.82 for a given airplane normal-force coefficient, the shock and the associated large negative pressures preceding the shock have moved from approximately 35 to 40 percent of the chord to about 60 to 65 percent of the chord (figs. 4 and 6(a) and 6(b)). The region of supersonic flow has extended beyond the midchord section at Mach numbers greater than 0.798 (fig. 7). The negative pressures over the midchord station have increased and those over the leading-edge section have decreased. The distributions over the rear 35 percent chord are essentially constant.

Mach numbers 0.82 to 0.89.– For a given normal-force coefficient, the shock remains at about the 55- to 60-percent-chord station for this Mach number range (figs. 4 and 6(b)). Gradual pressure fluctuations near the location of shock were noted. Within this range of Mach number, where the shock is approximately stationary, there is an increase in negative pressure coefficient over the airfoil ahead of the shock at Mach numbers of about 0.85 to 0.88 (figs. 6(a) and 6(b)). The more pronounced increases occur over the forward 20 percent chord and in the vicinity of the shock. The distributions over the section forward of the shock are similar, with the exception of the slight increase in negative pressures over the forward section as mentioned above. As the Mach number is increased above 0.82, an increase in negative pressures over the section trailing edge occurs, due to a region of separated flow behind the compression shock. At a Mach number of 0.872, about 65 percent of the upper surface is supersonic (fig. 7).

Mach numbers 0.89 to 0.95.– As shown in figures 4 and 6(b), the compression shock has moved from about the 65 percent chord to near the trailing edge with a corresponding increase in trailing-edge pressure coefficients. As the shock becomes approximately stationary at the trailing edge, local pressures are again increased. The general shape of the distribution forward of 60 percent chord remains approximately constant. At a free-stream Mach number of 0.90 (fig. 7) supersonic flow exists over the entire upper surface, with exception of the leading edge and possibly the trailing edge, and maximum local Mach numbers of about 1.4 are attained at 65 to 70 percent of the chord. The maximum local Mach numbers obtained in this region were about 1.5 for normal-force coefficients of about 0.33 and occurred over 75 to 95 percent of the section at a free-stream Mach number of 0.947.

Mach numbers 0.95 to 1.09.– As the Mach number is increased from 0.95 to 1.09, the shape of the distributions over the upper surface remains approximately the same but the magnitudes of loads over the upper surface are progressively reduced (figs. 4 and 6(b)).

Mach numbers 1.14 to 1.25.– The form of the load distribution over the upper surface for this Mach number range (fig. 5) is similar to that presented for Mach numbers of 0.95 to 1.09 in figure 4. The magnitude of pressures over the surface increases positively by 40 to 50 percent as the Mach number is increased from 1.14 to 1.25. For an increase in section normal-force coefficient from 0.19 to 0.93 at a Mach number of about 1.16 the upper-surface distribution approaches a rectangular shape, and a maximum local Mach number of about 2.0 is reached over the trailing edge of the airfoil at a C_n of 0.93. (See fig. 9.) There was no indication that maximum lift had been reached at this normal-force coefficient. The shape of the lower-surface distribution remained essentially the same.

Lower-Surface Pressure Distribution

As shown in figures 4 and 5, the same general change in load distribution is shown to occur over the lower surface as shown over the upper surface; that is, with the formation of shock and increase in Mach number, there is a shifting of the load center toward the rear. However, at the lower Mach numbers the existing flow over the forward section of the lower surface presents a more favorable pressure gradient than is present over the upper surface. The shock movement over the lower surface to the rear is more uniform and more rapid than that over the upper surface. These facts are pointed out in the following discussion for selected Mach number ranges.

Mach numbers 0.75 to 0.82.– From figures 4 and 6(c) and 6(d), it is shown that there is a slight increase negatively in pressures over the forward 50 percent chord and that the load over the rear 50 percent remains essentially constant. There is no definite indication of shock formation at these Mach numbers and normal-force coefficients.

Mach numbers 0.82 to 0.89.– Shock formation is indicated after sonic velocity is attained at about 55 percent of the chord near a Mach number of 0.84 and moves to about 80 percent chord at a Mach number of 0.89. The shock movement results in an increase in pressures positively forward of the shock and a marked increase in trailing-edge loads. The region of supersonic flow extends from about 45 to 85 percent of the chord at a Mach number of 0.89.

Mach numbers 0.89 to 0.95.– The shock continues to move rearward to about 90 percent of the chord at a Mach number of 0.92 and, from the data obtained in these tests, the shock appears to remain at this location. The formation of a stationary shock wave in this vicinity may be due to the reflexed trailing edge of the airfoil. The region of supersonic flow extends over about the rear 75 percent of the chord.

Mach numbers 0.95 to 1.23.– The general shape of the distribution of load over the surface remains essentially the same above a Mach number of 0.95 and, with the exception of the leading edge and trailing edge, is similar to the distribution of the upper surface. With an increase in Mach number for a given normal-force coefficient, there is a slight positive increase in the pressure coefficient over the surface.

Total Section Loads

A gradual transition of load distribution from one having peak loads near the leading edge to one approximating a rectangular shape occurs when the Mach number is increased from the subsonic to the supersonic values. (See fig. 10.) The total load distributions are irregular in shape for Mach numbers up to 0.95 due to variations of the shock location and movement over the upper and lower surfaces. The shapes of the distributions at Mach numbers greater than 0.95 are essentially the same since the shock waves on both surfaces remain near the trailing edge.

Due to the variation of the shock position on one surface with respect to the other and to the different rates of shock movement on each surface, there is a large shifting of the section load center and a corresponding large change in section pitching moment. The center of load at a Mach number of 0.75 (fig. 11) is located at approximately 26 percent of the section chord and moves to about 42 percent chord as the Mach number is increased from 0.75 to 0.85 due to the upper-surface shock movement to the rear. For the same Mach number range the pitching-moment coefficient increases from −0.009 to −0.067. As the Mach number is increased from 0.85 to 0.89, the center of load moves forward to approximately 26 percent chord due to the rearward shift of the lower-surface shock and to the approximately stationary upper-surface shock (fig. 8). At a Mach number of about 0.90, where the shock on the lower surface has almost reached the trailing edge and the shock on the upper surface moves rearward, the center of load moves rapidly to about 45 percent chord at a Mach number of 0.95. Due to a more rapid reduction of negative pressures over the forward section of the upper surface than over the rear section at Mach numbers above 1.00, there is a gradually rearward shift of the center of load to about 51 percent chord at Mach numbers of 1.25. Due to the large and fairly rapid changes in the wing section pitching-moment coefficient between Mach numbers of 0.75 and 1.0 it may be expected that large trim changes would occur in this Mach number range. Correspondingly high load factors may result in flights at low altitudes. The occurrence of high load factors would depend however on the rate at which the airplane traversed the Mach number range of 0.75 to 1.0 and on the rate of application and magnitude of trimming forces that may be available to the pilot. Because of the increased trailing-edge loads, large changes in section hinge moments occur in a Mach number range of 0.80 to 0.96 (fig. 11). The ratio of the section normal-force coefficient to the airplane normal-force coefficient (fig. 11) shows that the section load abruptly increases at a Mach number of 0.95 and remains at about this load level as the Mach number is increased to 1.25. The decrease in section load at a Mach number of 0.75 indicated in figure 11 is based on one load distribution and no explanation of this occurrence can be made at this time. However, total wing loads measured by strain gages located at the wing root indicate that the lateral center of load remains essentially constant within this Mach number range.

CONCLUSIONS

Results of the chordwise pressure-distribution measurements over the wing of the XS-1 airplane show that:

1. As the Mach number is increased from 0.75 to 0.95 for a normal-force coefficient of about 0.33, there is a rearward shift of the surface load center on both the upper and lower surfaces.

2. At free-stream Mach numbers from 0.95 to 1.25 where the shock is located near the trailing edge, the distributions of load over the two surfaces are somwhat similar at comparable normal-force coefficients. At high normal-force coefficients at a Mach number of 1.16, however, the upper-surface pressure distribution approaches a rectangular distribution.

3. The maximum local Mach number reached in these tests was about 2.0 at a free-stream Mach number of about 1.16 and a section normal-force coefficient of 0.93.

4. Large trim changes of the airplane may occur in a Mach number range of 0.80 and 1.0 due to the changes in wing pitching moments. Corresponding high load factors may result during flights at low altitude.

5. The section total load distribution is transformed from one having peak loads near the leading edge at subsonic speeds to one approaching a rectangular distribution at supersonic speeds. The distribution over the section for Mach numbers of 0.95 to 1.19 are similar in shape.

6. There are large changes in section hinge moments due to the large changes in trailing-edge loads.

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MLE

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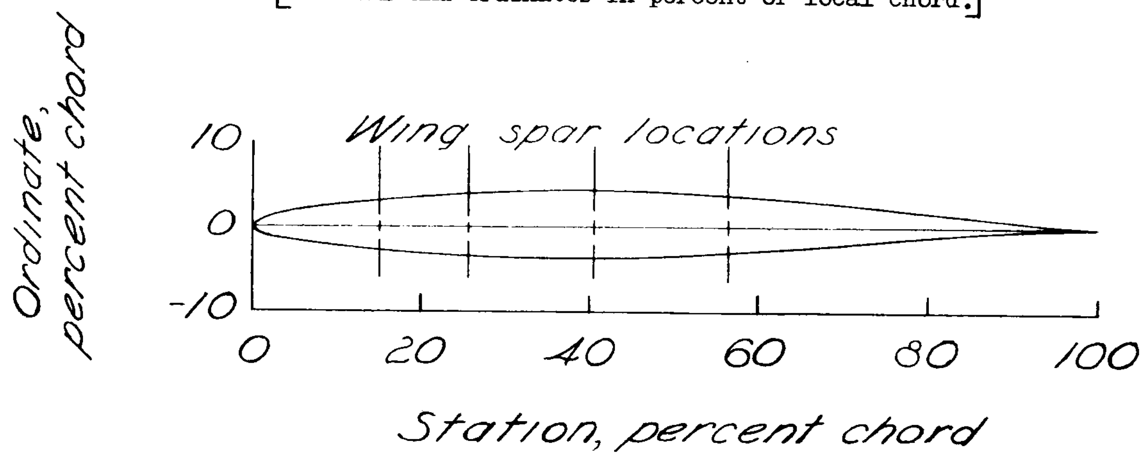
1. Drake, Hubert M., McLaughlin, Milton D., and Goodman, Harold R.: Results Obtained during Accelerated Transonic Tests of the Bell XS-1 Airplane in Flights to a Mach Number of 0.92. NACA RM No. L8A05a, 1948.

2. Williams, W. C., and Beeler, De E.: Results of Preliminary Flight Tests of the XS-1 Airplane (8-Percent Wing) to a Mach Number of 1.25. NACA RM No. L8A23a, 1948.

TABLE I.– PROFILE AND ORDINATES OF THE AIRFOIL TEST SECTION

[Stations and ordinates in percent of local chord.]



Station	Upper surface	Lower surface
0	0	0
.50	.637	–.597
.75	.773	–.717
1.25	.978	–.892
2.50	1.334	–1.185
5.00	1.868	–1.613
7.50	2.285	–1.946
10.00	2.636	–2.225
15.00	3.200	–2.662
20.00	3.635	–2.999
25.00	3.962	–3.245
30.00	4.190	–3.420
35.00	4.350	–3.528
40.00	4.425	–3.570
45.00	4.410	–3.535
50.00	4.290	–3.410
55.00	4.060	–3.185
60.00	3.745	–2.886
65.00	3.355	–2.535
70.00	2.915	–2.137
75.00	2.425	–1.708
80.00	1.910	–1.270
85.00	1.378	–.838
90.00	.854	–.442
95.00	.371	–.118
100.00	0	0
Slope of radius through L.E.: 0.038875.		

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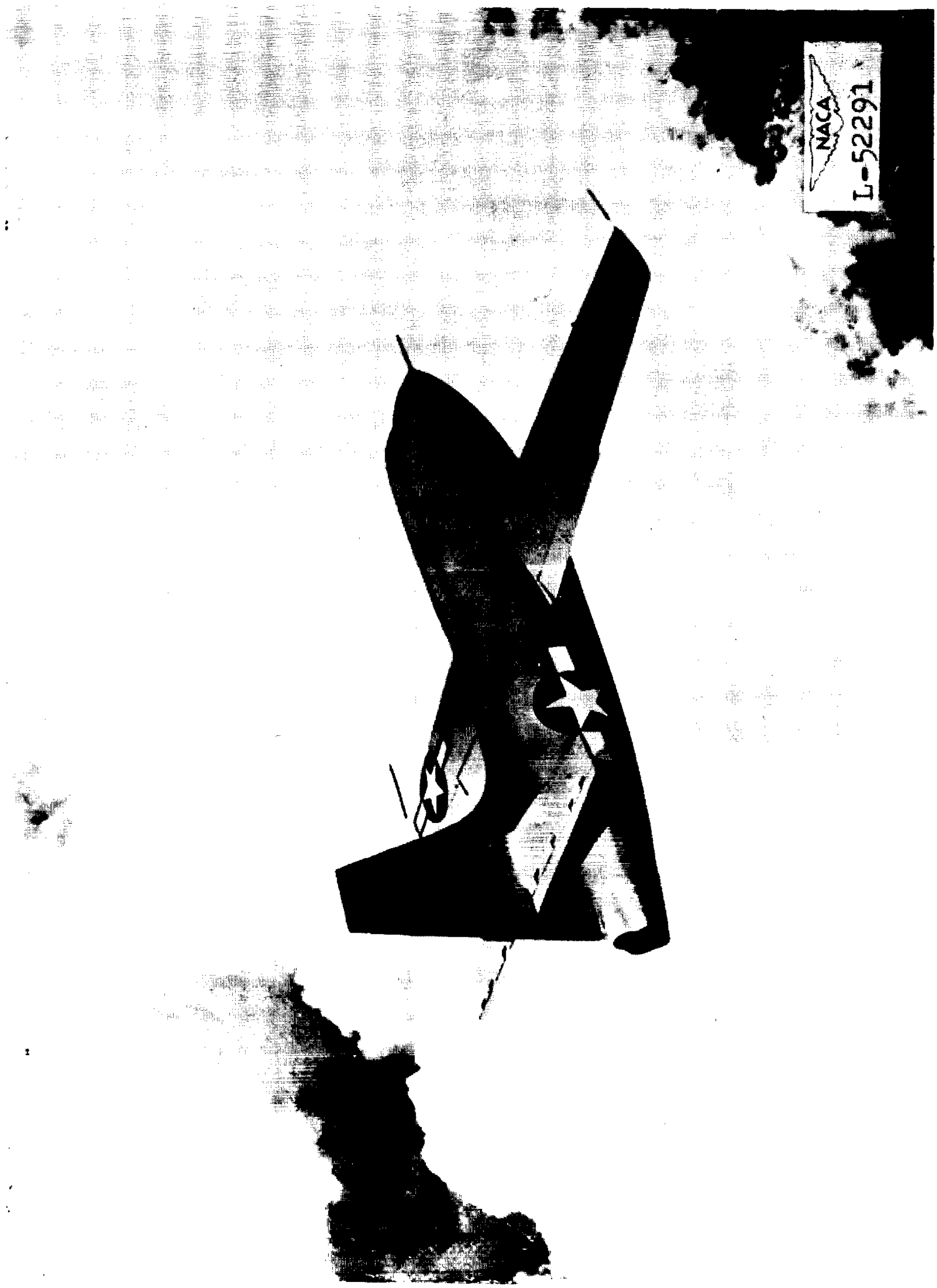


Figure 1.- Photograph of XS-1 airplane.

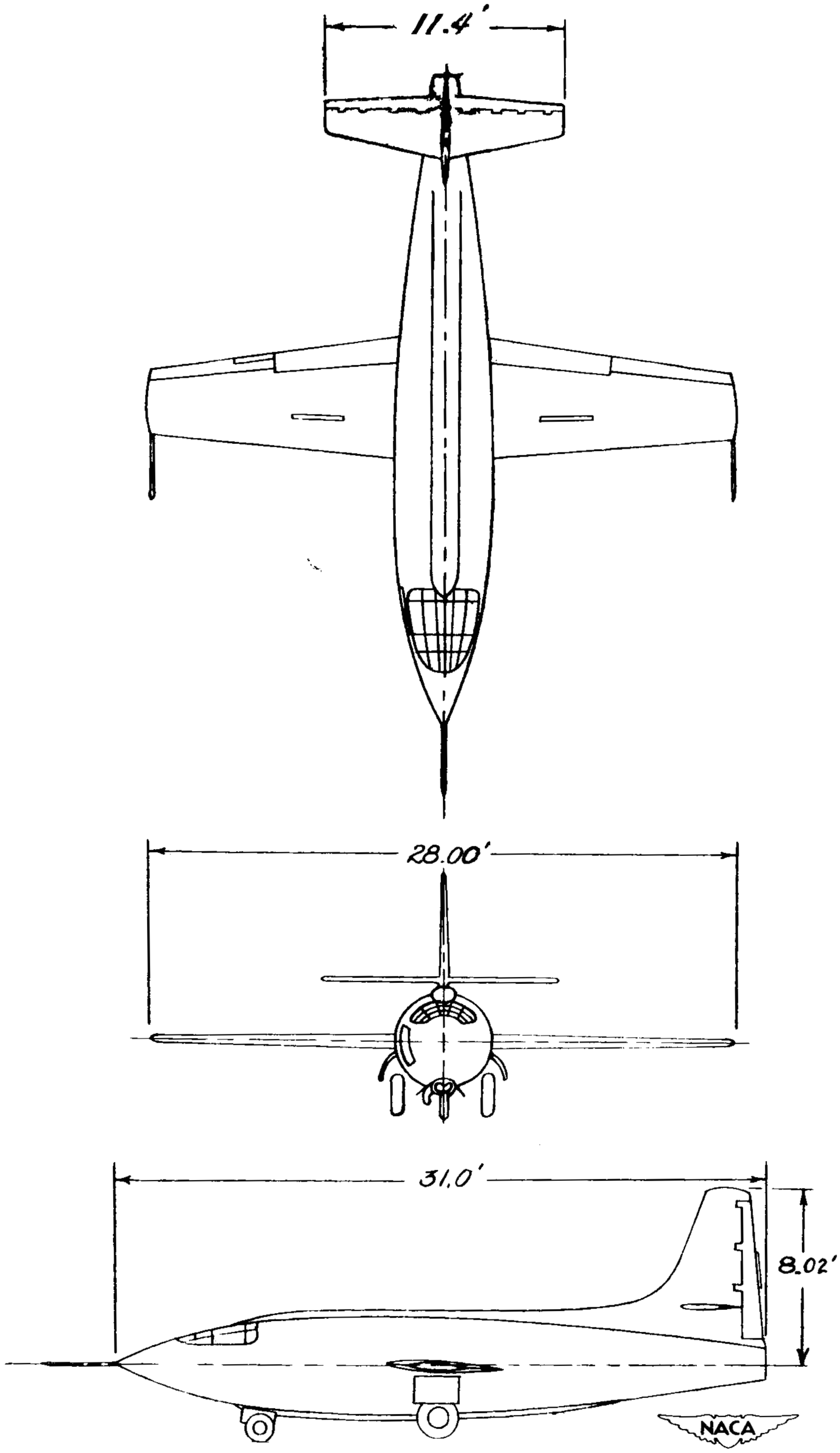
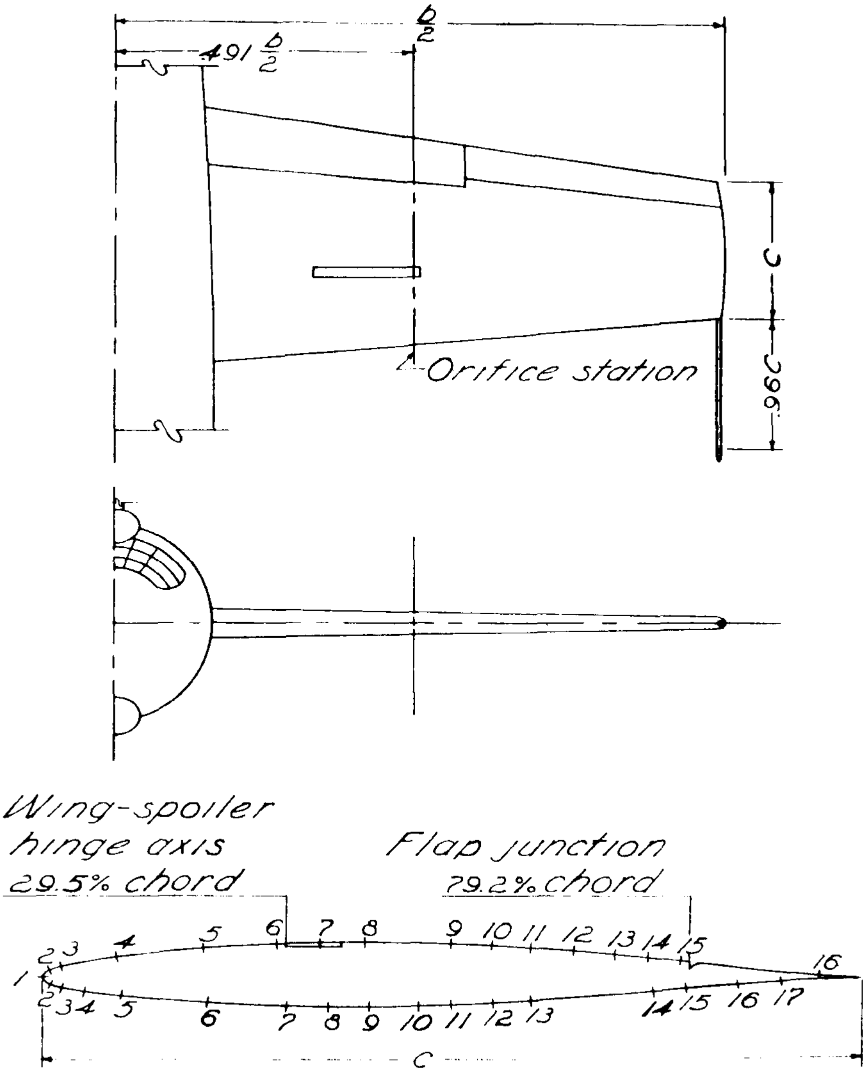


Figure 2.- Three-view drawing of XS-1 airplane.



UPPER SURFACE

Orifice No.	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16
Station, percent c	0	1.25	2.5	9.5	20.0	28.9	34.4	39.4	50.1	55.1	59.9	65.2	70.1	74.2	78.2	94.9

LOWER SURFACE

Orifice No.	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15	16	17
Station, percent c	0	1.25	2.5	5.1	9.6	20.2	30.1	35.1	40.1	46.0	50.1	55.1	60.2	74.7	78.4	84.9	90.0

Figure 3.– Location of pressure-measuring stations for obtaining wing pressure distribution.

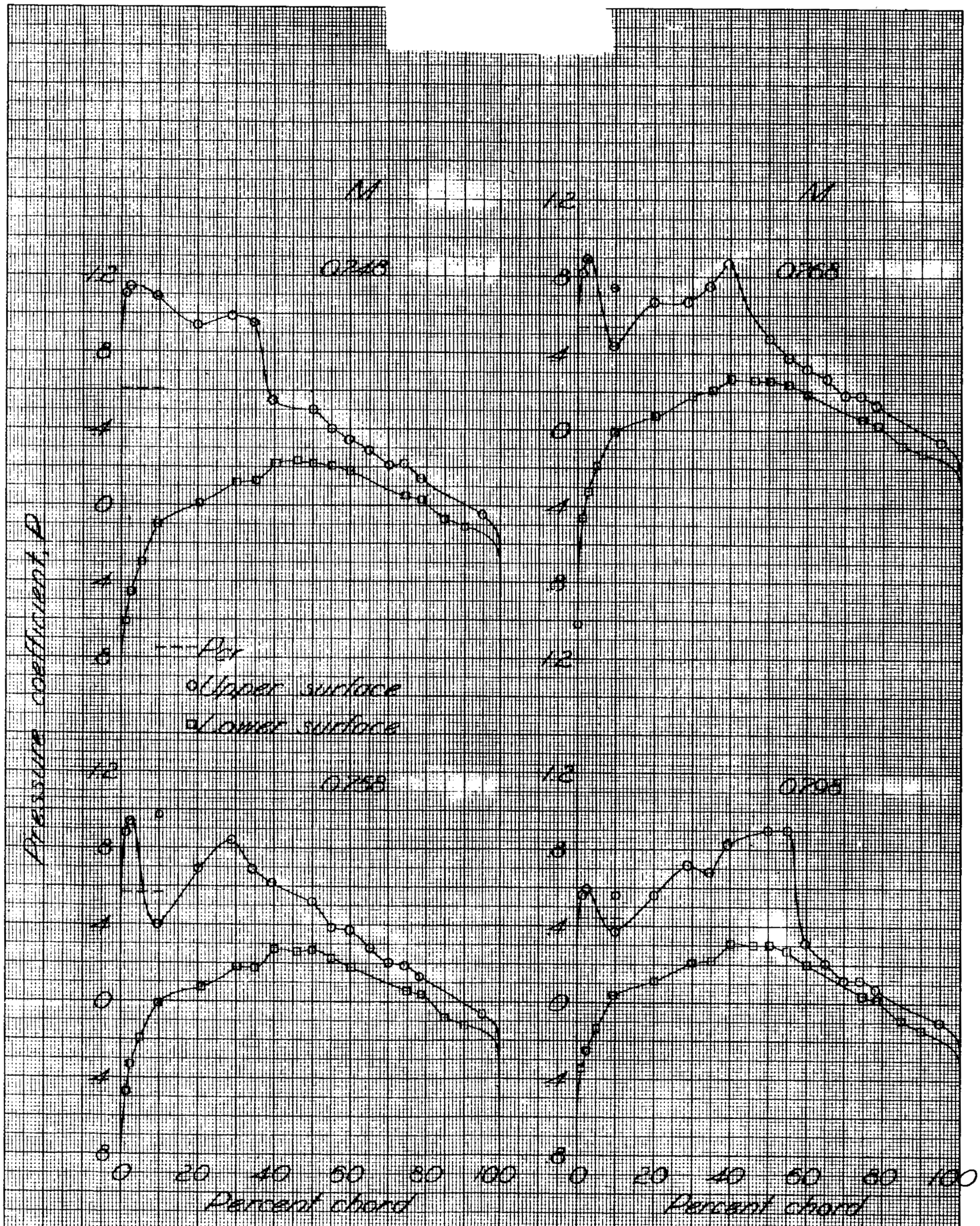


Figure 4.- Pressure distribution over a midspan-wing station of XS-1 airplane. Wing thickness, .08 chord. C_{N_A} = .33±.00.

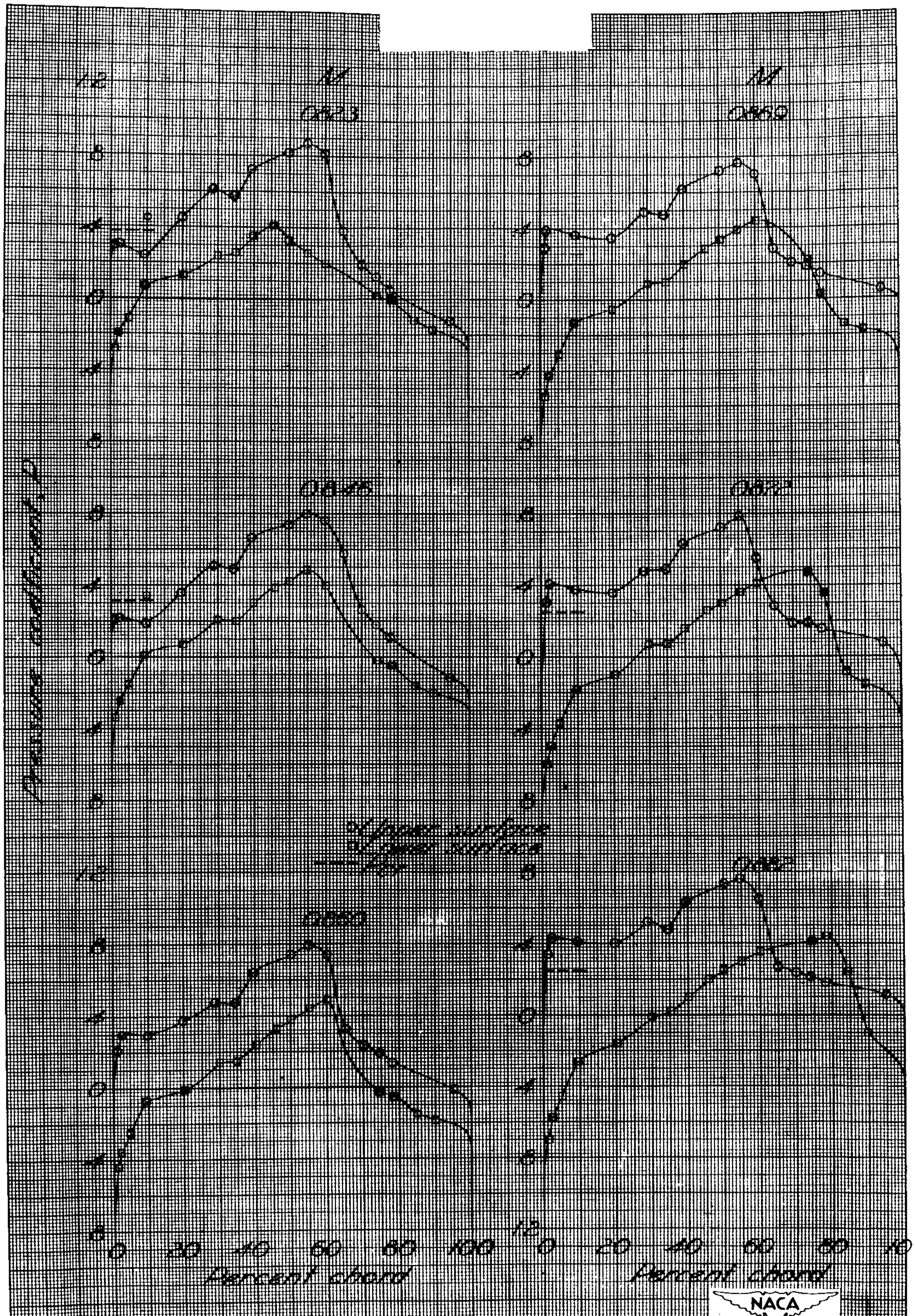


Figure 4.- Continued.

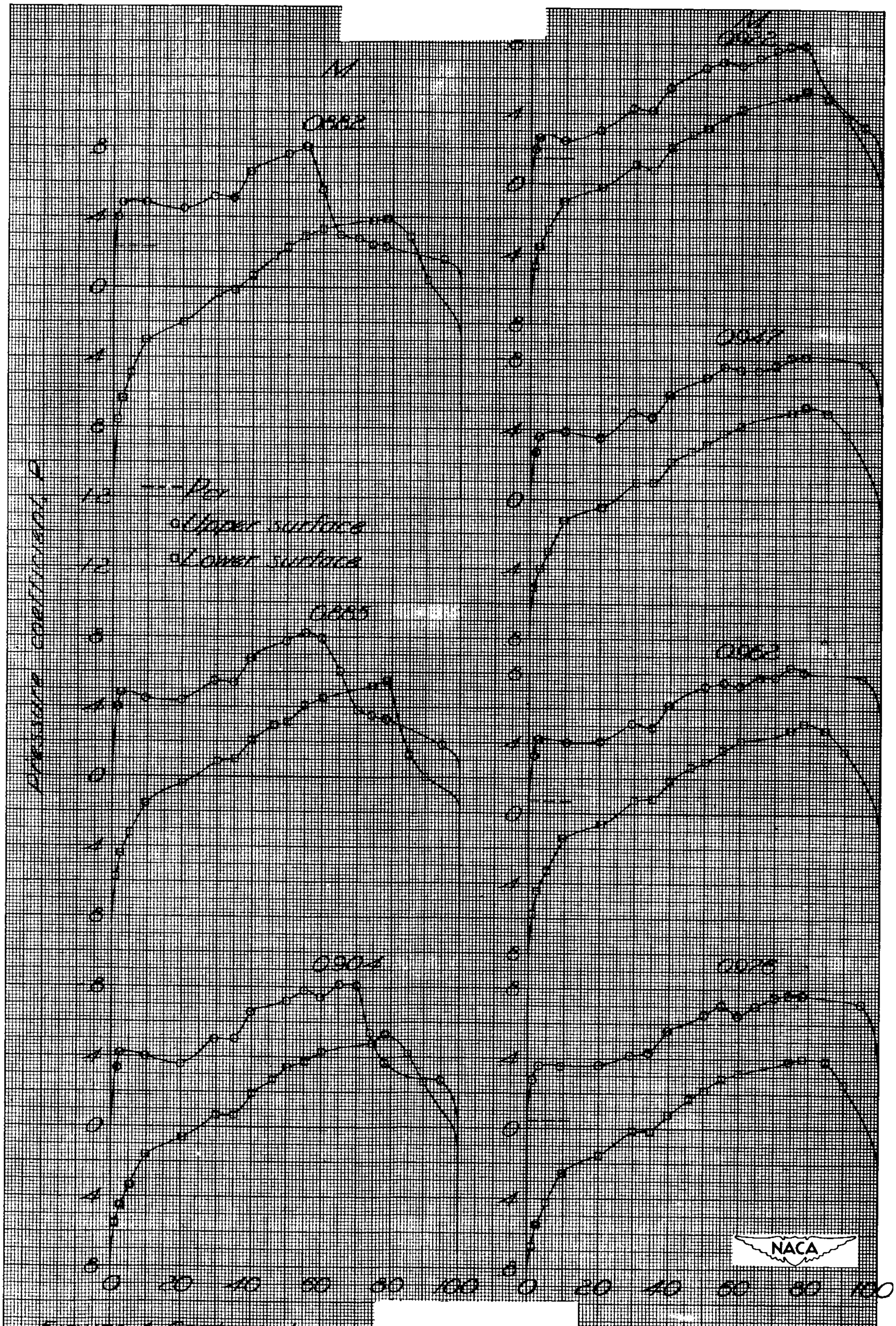


Figure 4.-Continued.

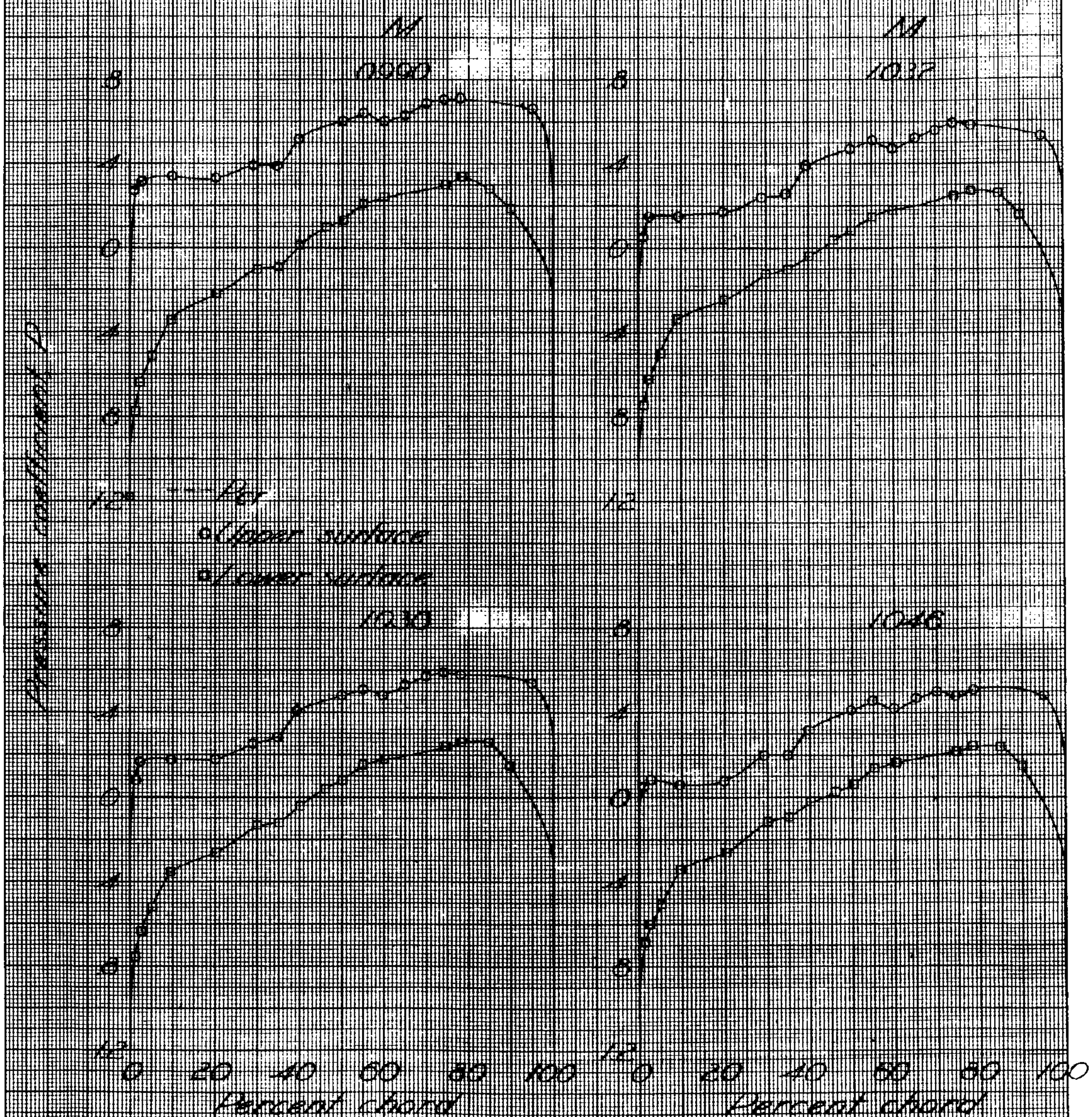


Figure 4 Continued.

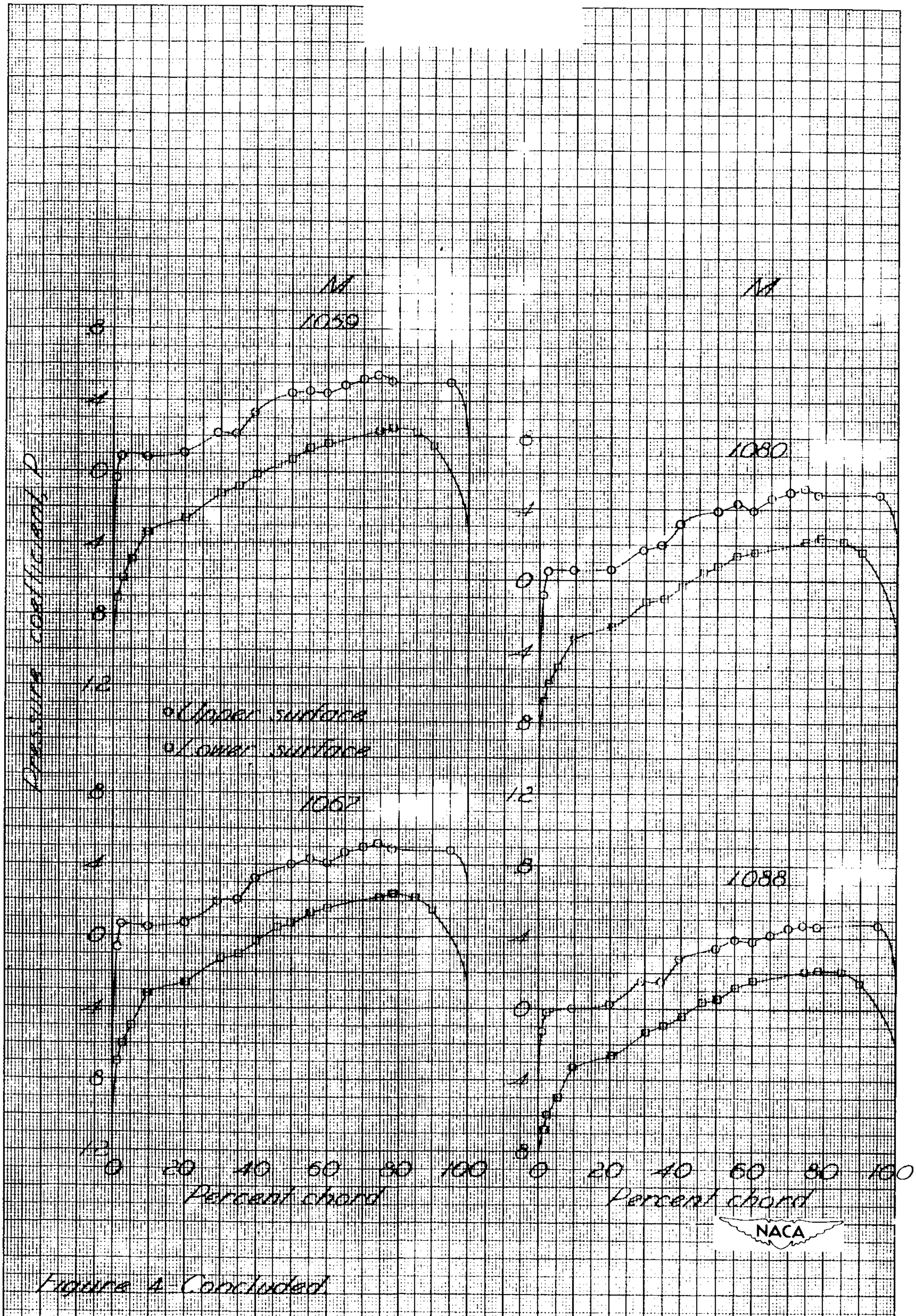


Figure 4. Concluded.

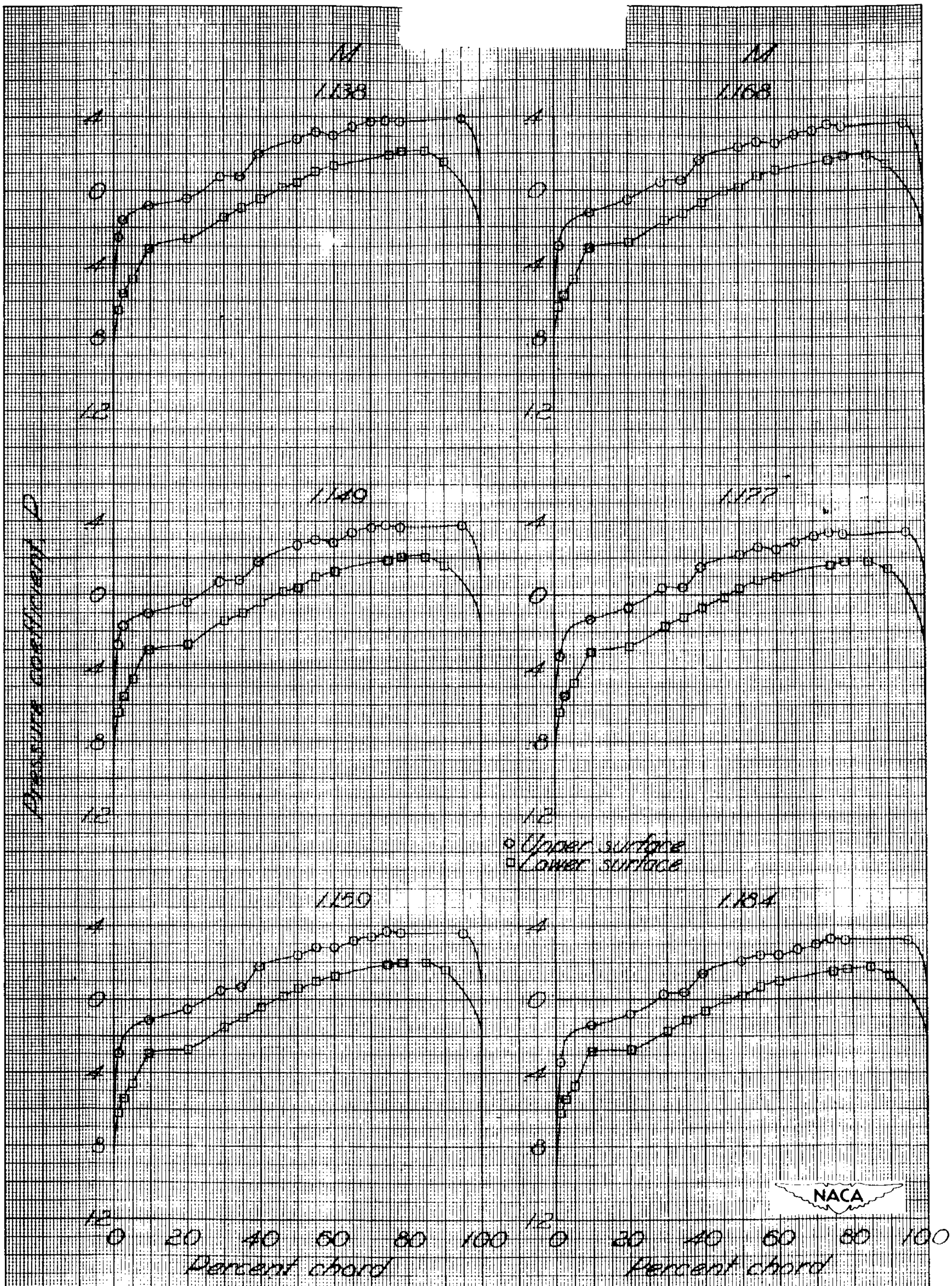


Figure 5. Pressure distribution over a midspan-wing station of XS-1 airplane. Wing thickness, .08 chord. C_{N_A}, .16±0.4.

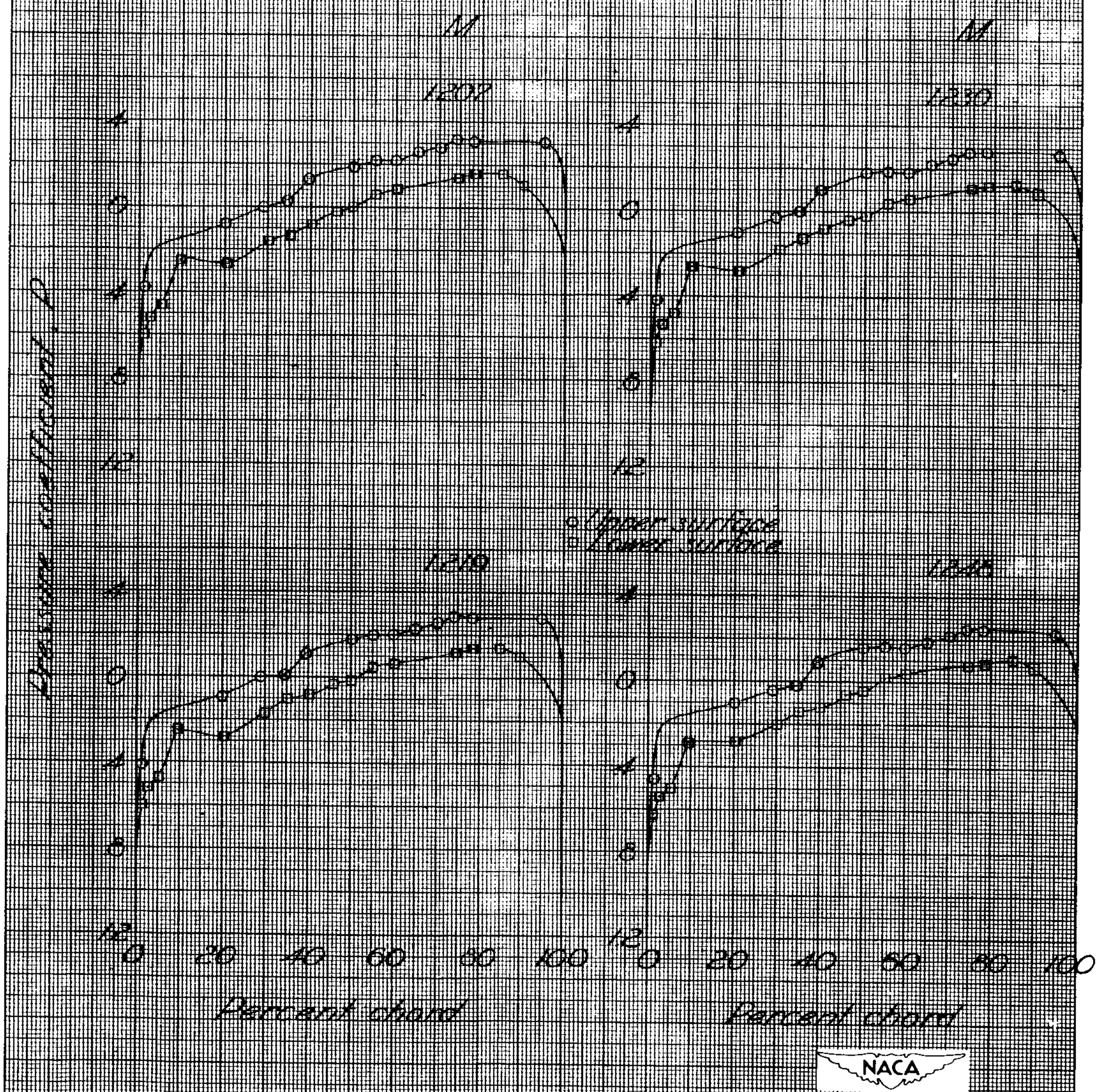
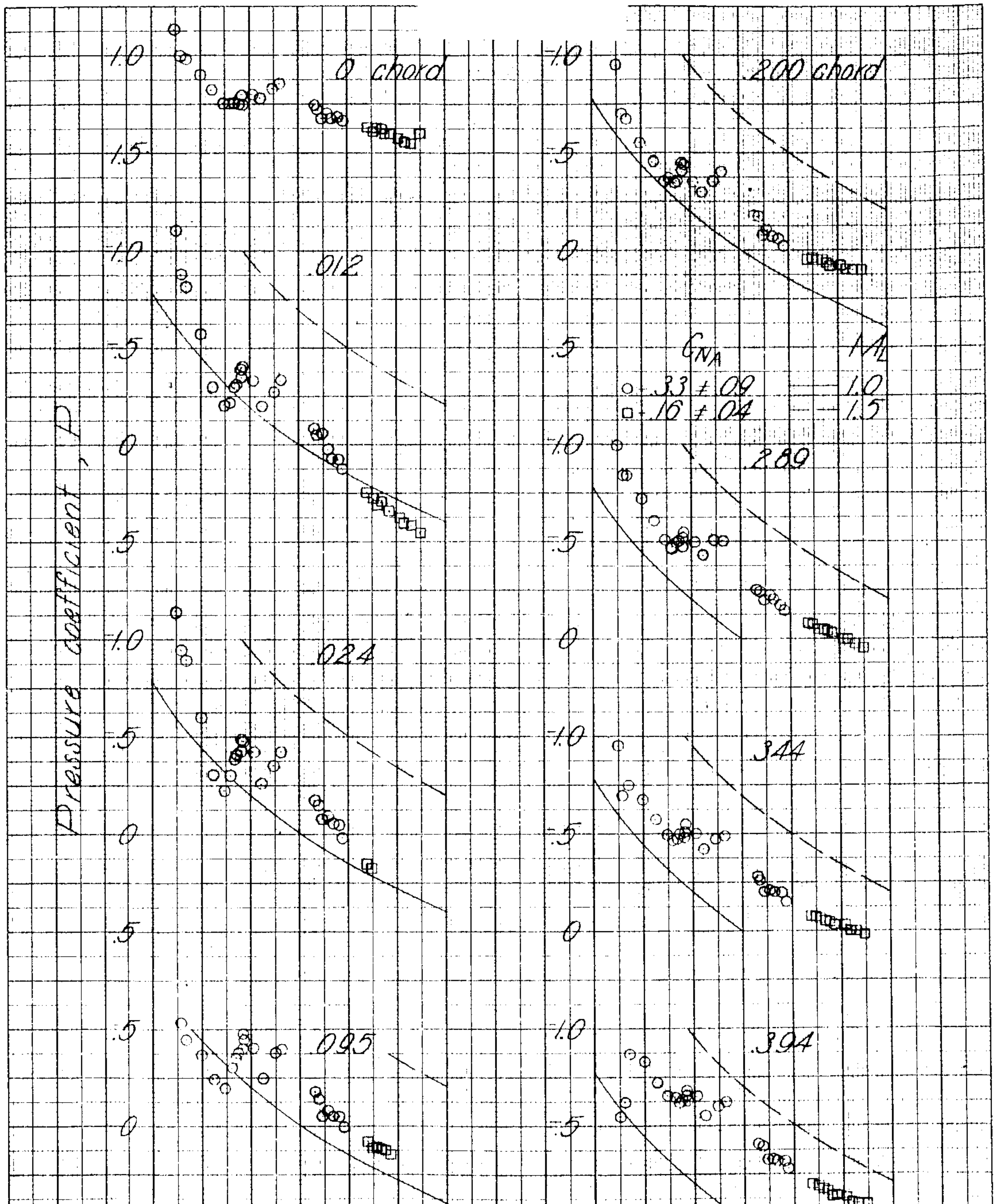
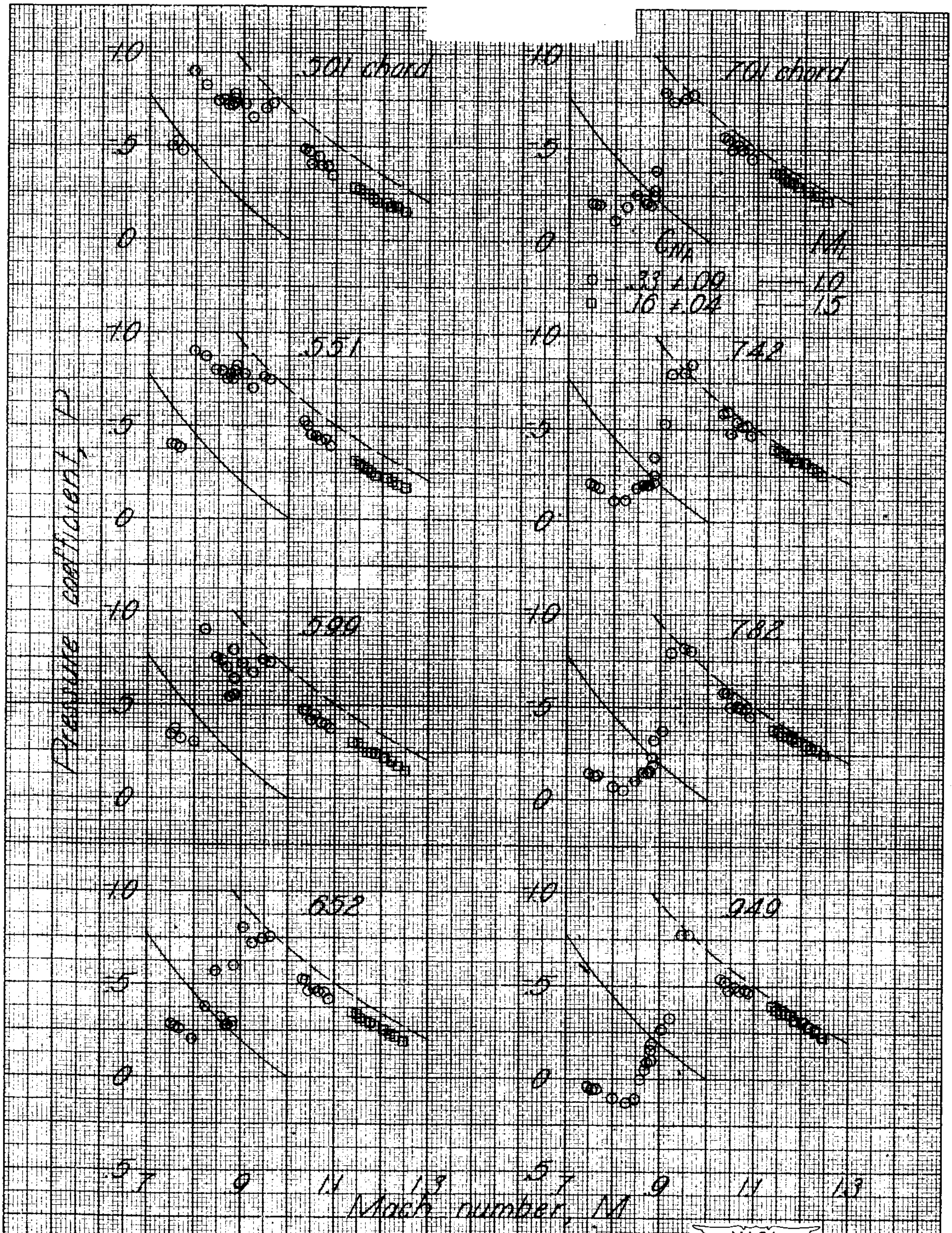


Figure 5. Concluded.



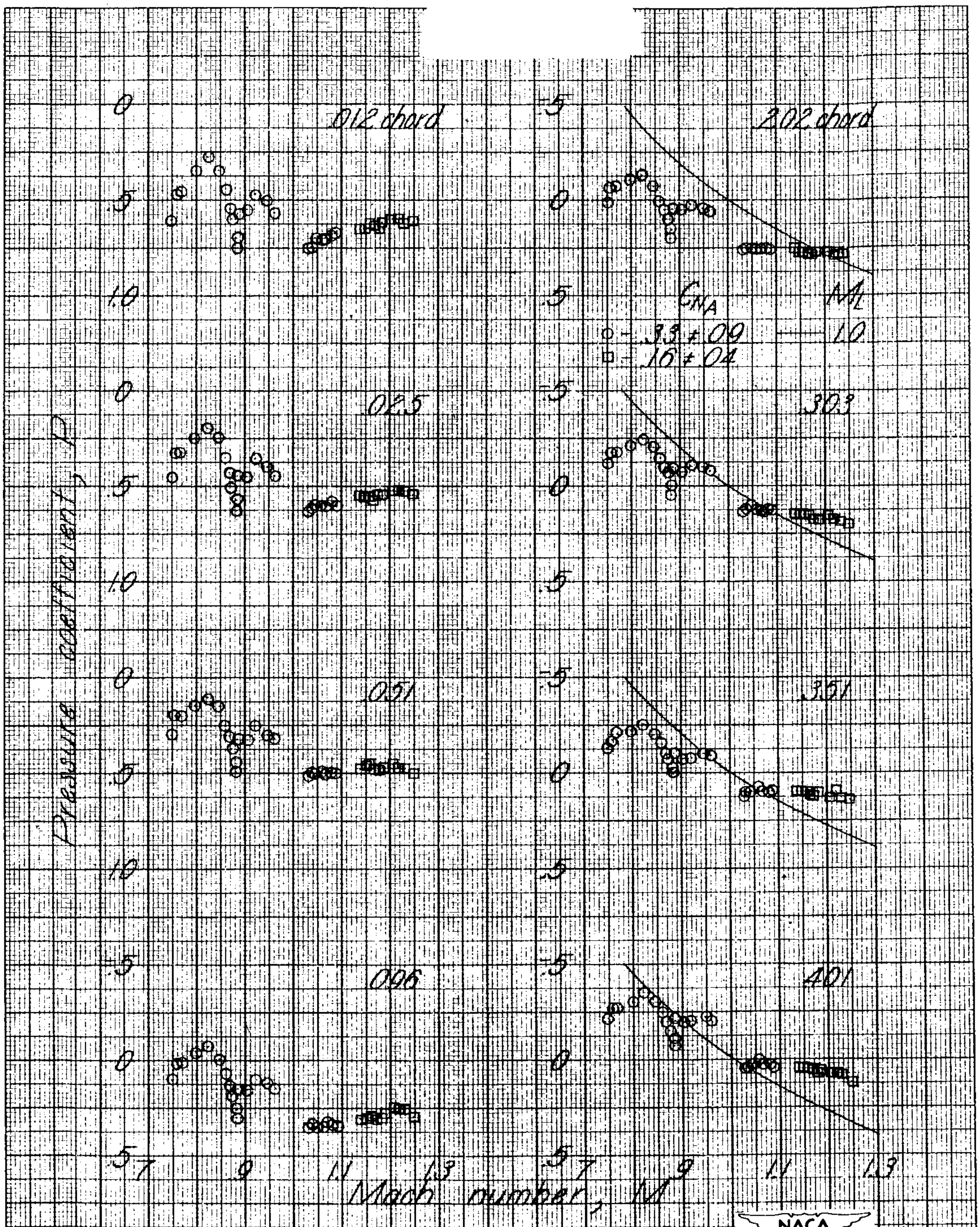
(a) Upper surface.

Figure 6.- Variation of local pressure coefficient with free-stream Mach number. Midspan-wing station of the XS-1 airplane. Wing thickness, .08 chord.



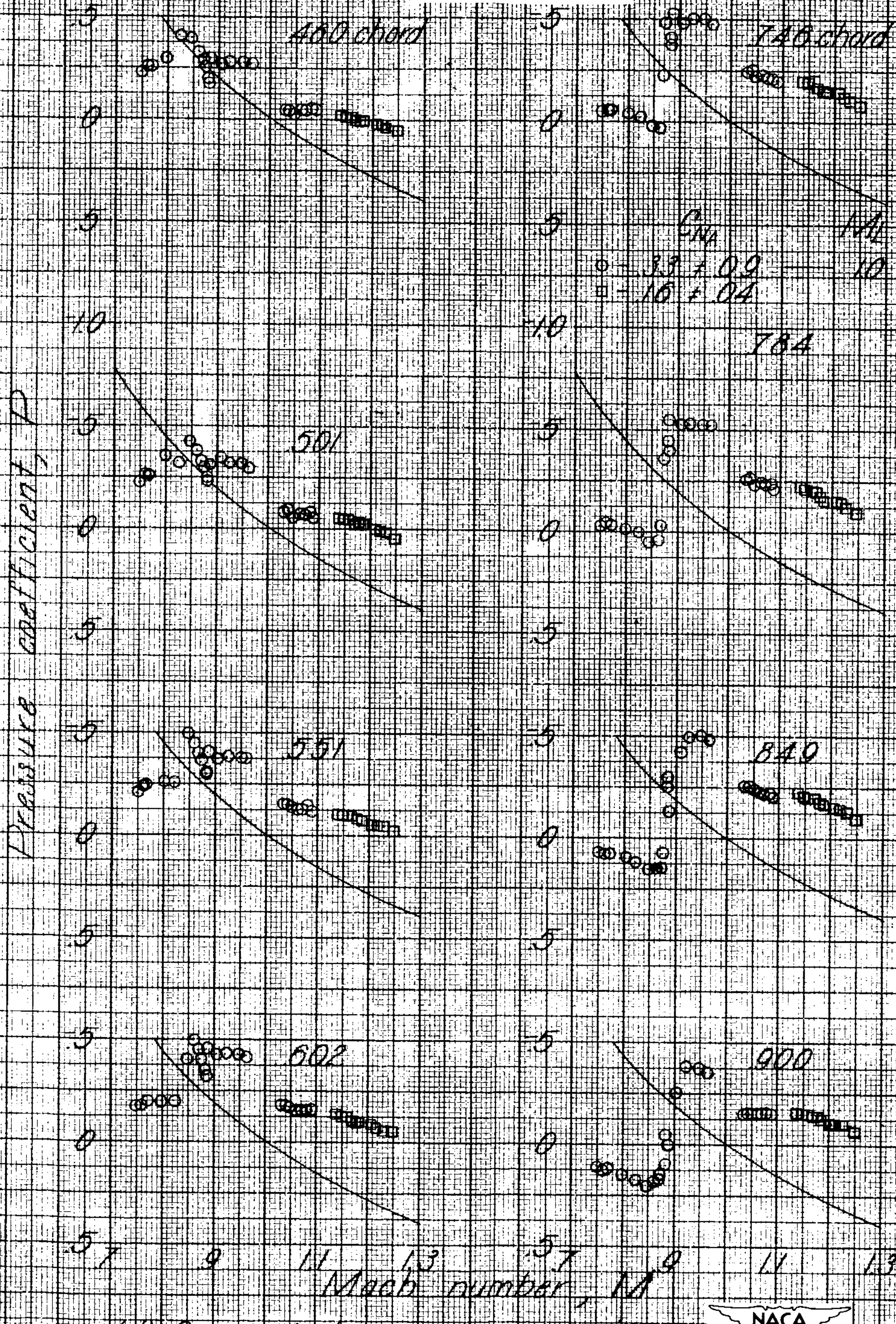
(b) Concluded.

Figure 6.- Continued.



(c) Lower surface.

Figure 6.- Continued.



(d) Concluded.

Figure 6 - Concluded.

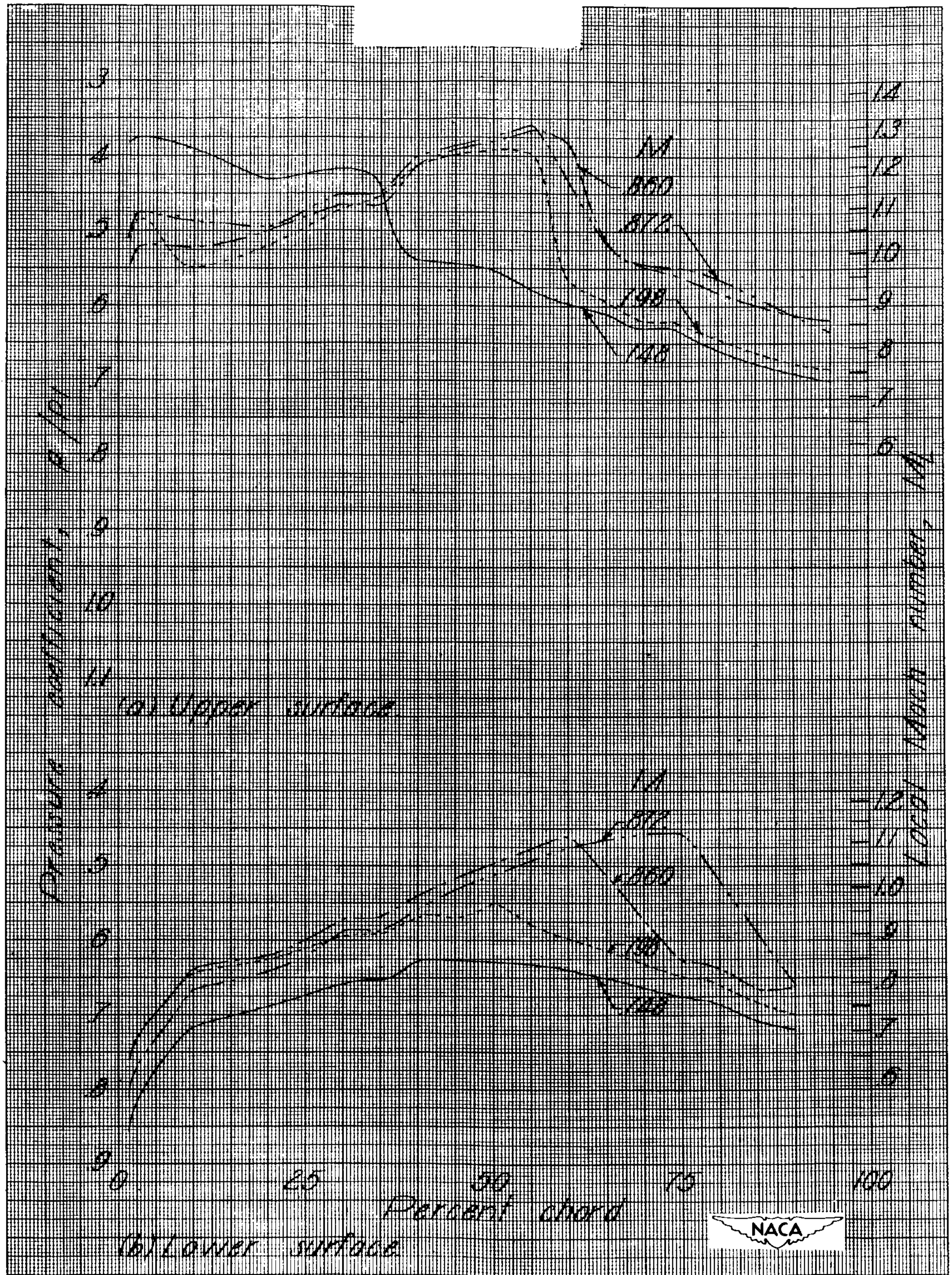


Figure 7.- Distributions showing local velocity regions over individual surfaces.

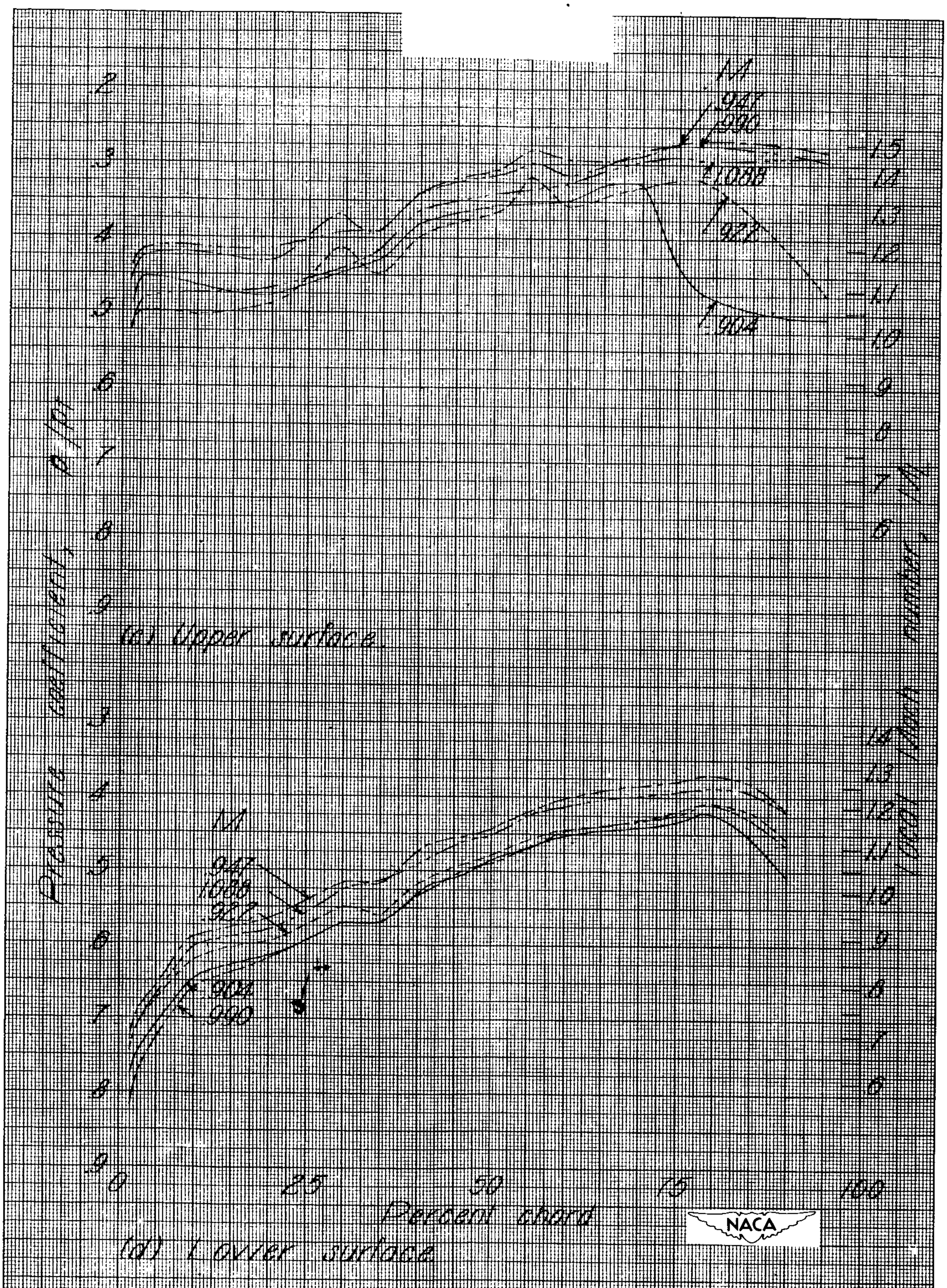


Figure 7.- Concluded.

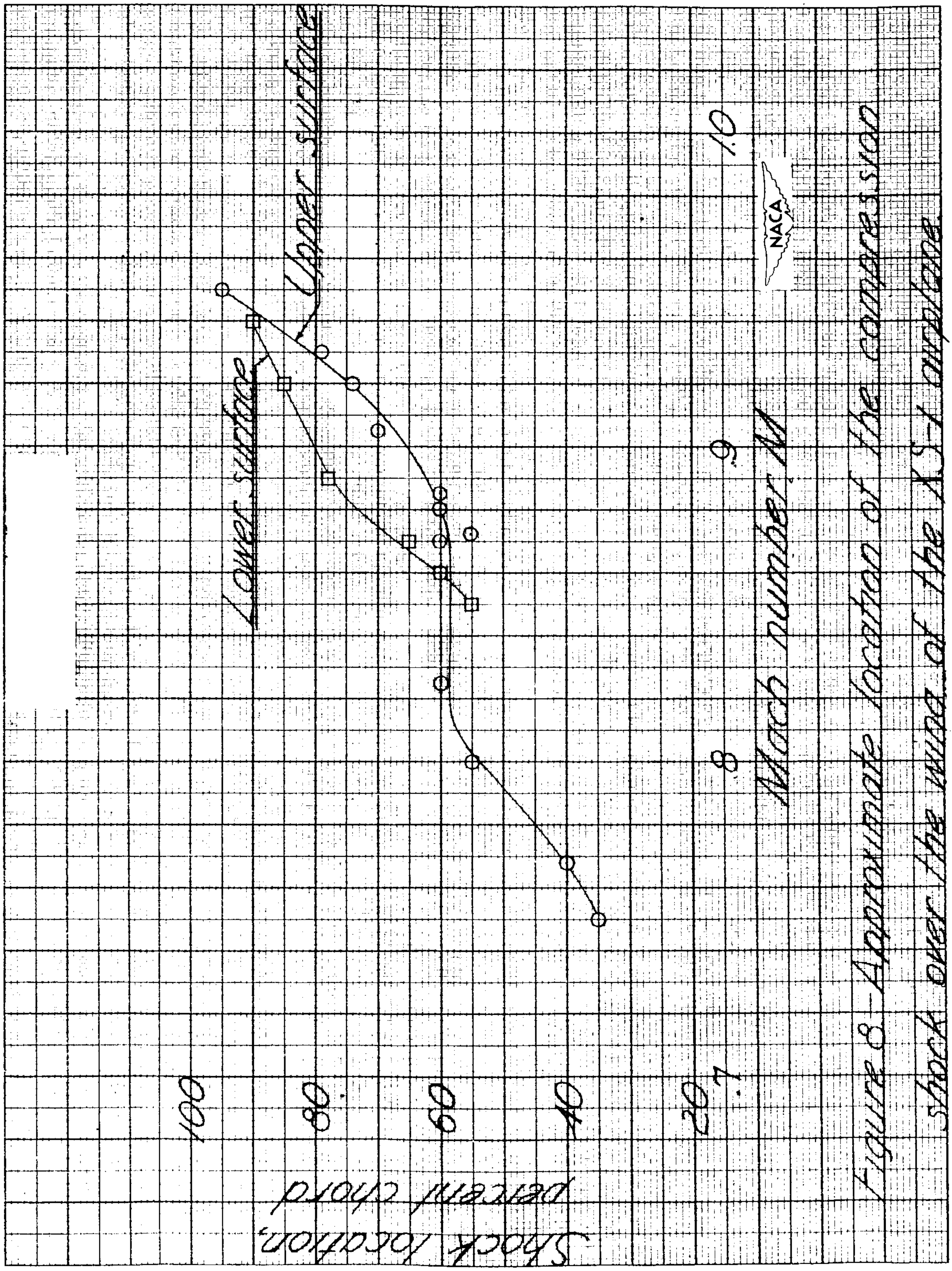


Figure 8.- Approximate location of the compression shock over the wing of the XS-1 airplane.

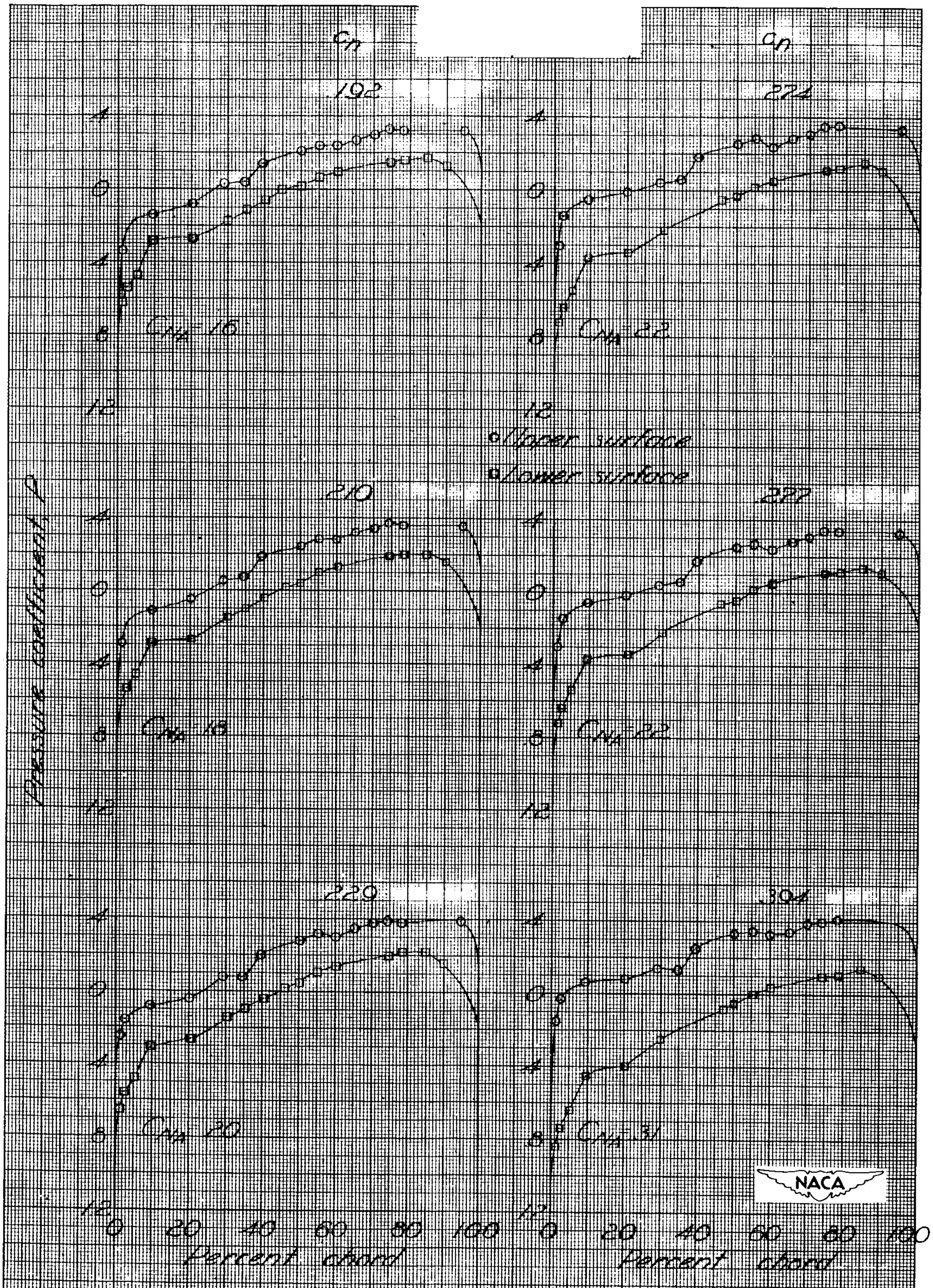


Figure 9.- Pressure distribution over a midspan-wing station of XS-1 airplane. Wing thickness, 0.8 chord. M 1.16-0.4.

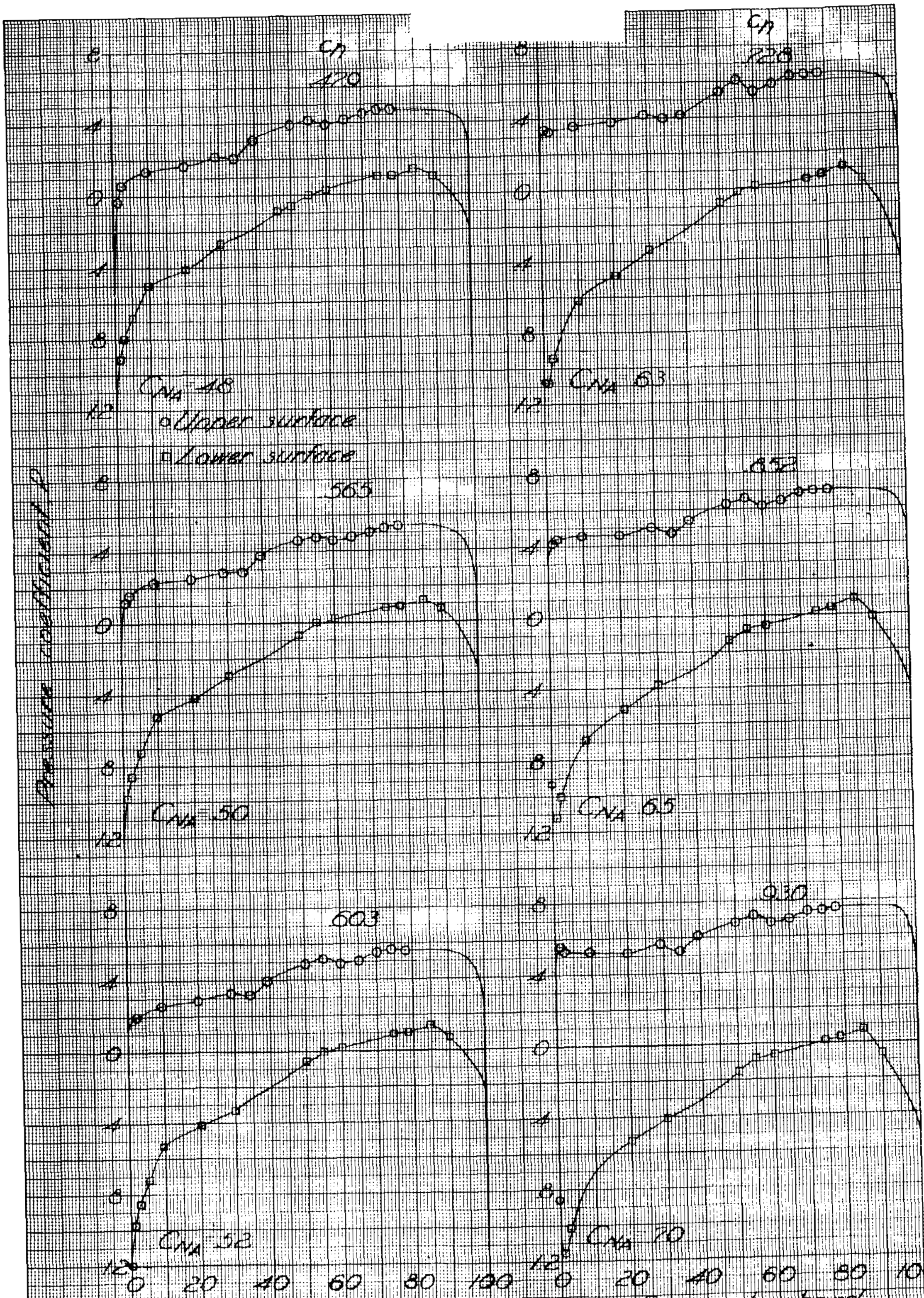


Figure 9- Concluded.

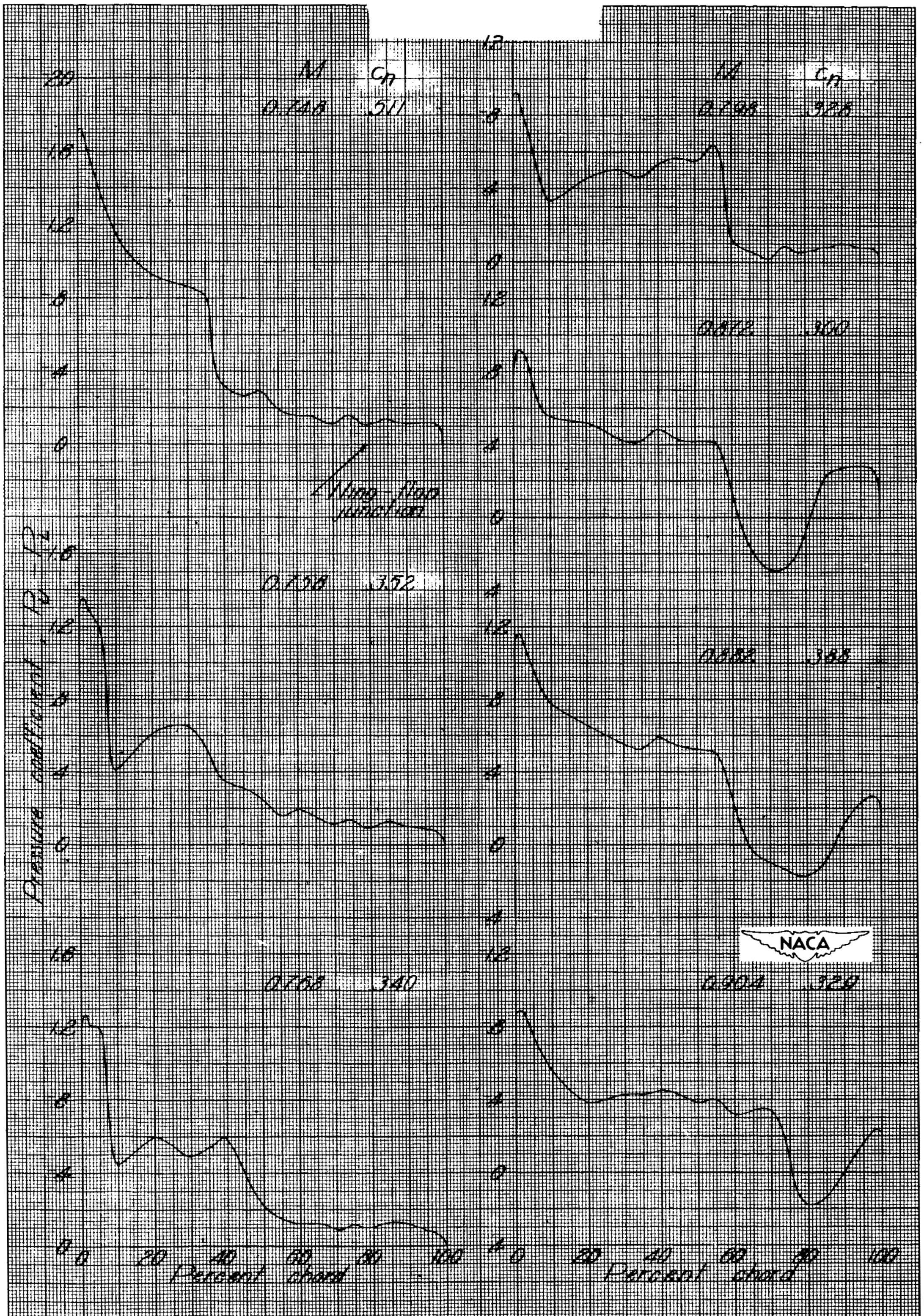


Figure 10.- Chordwise loading of a midspan wing station of XS-1 airplane. Wing thickness, 0.08 chord. $C_{N_A} = 0.35 \pm .08$

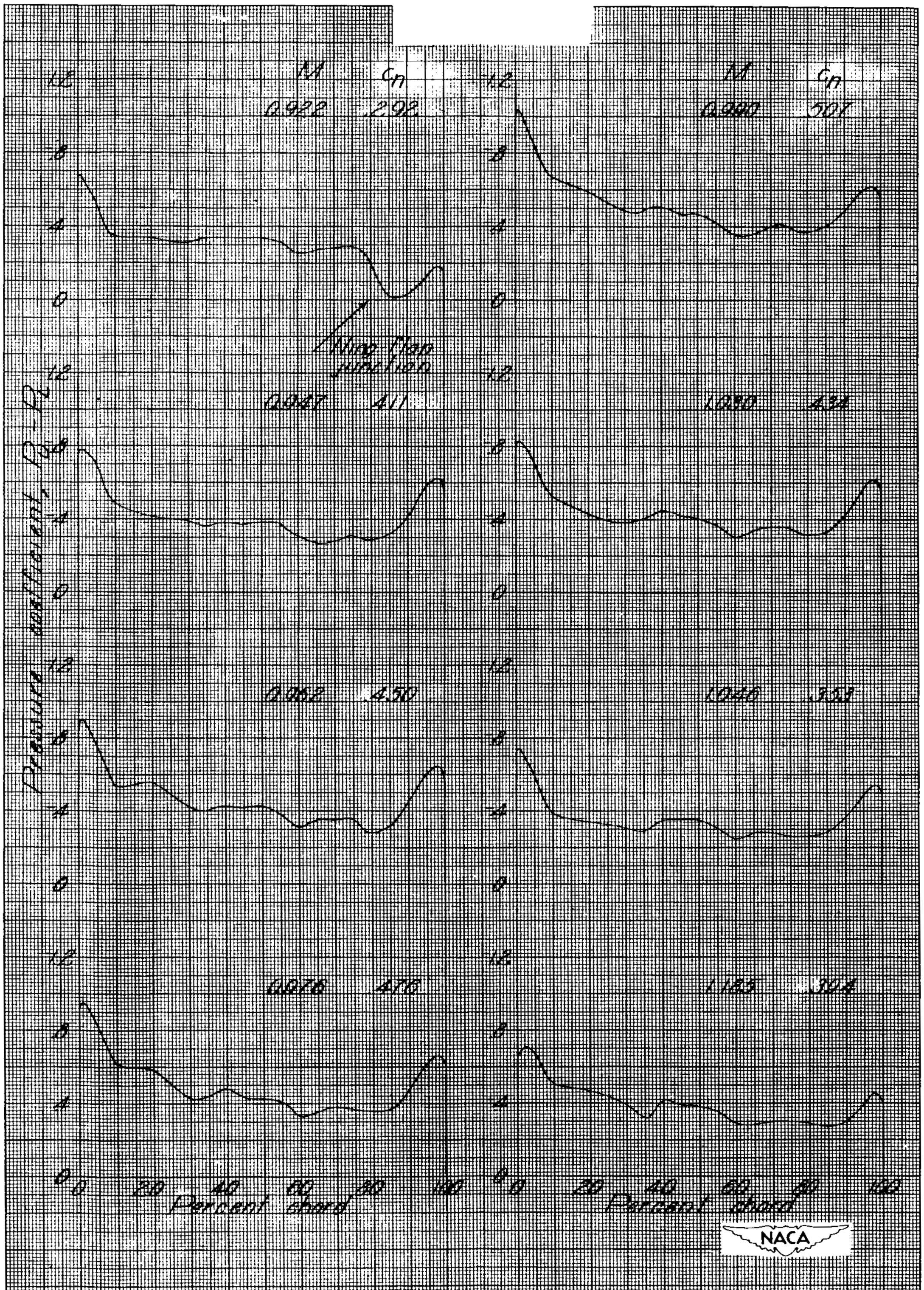


Figure 10.- Concluded.

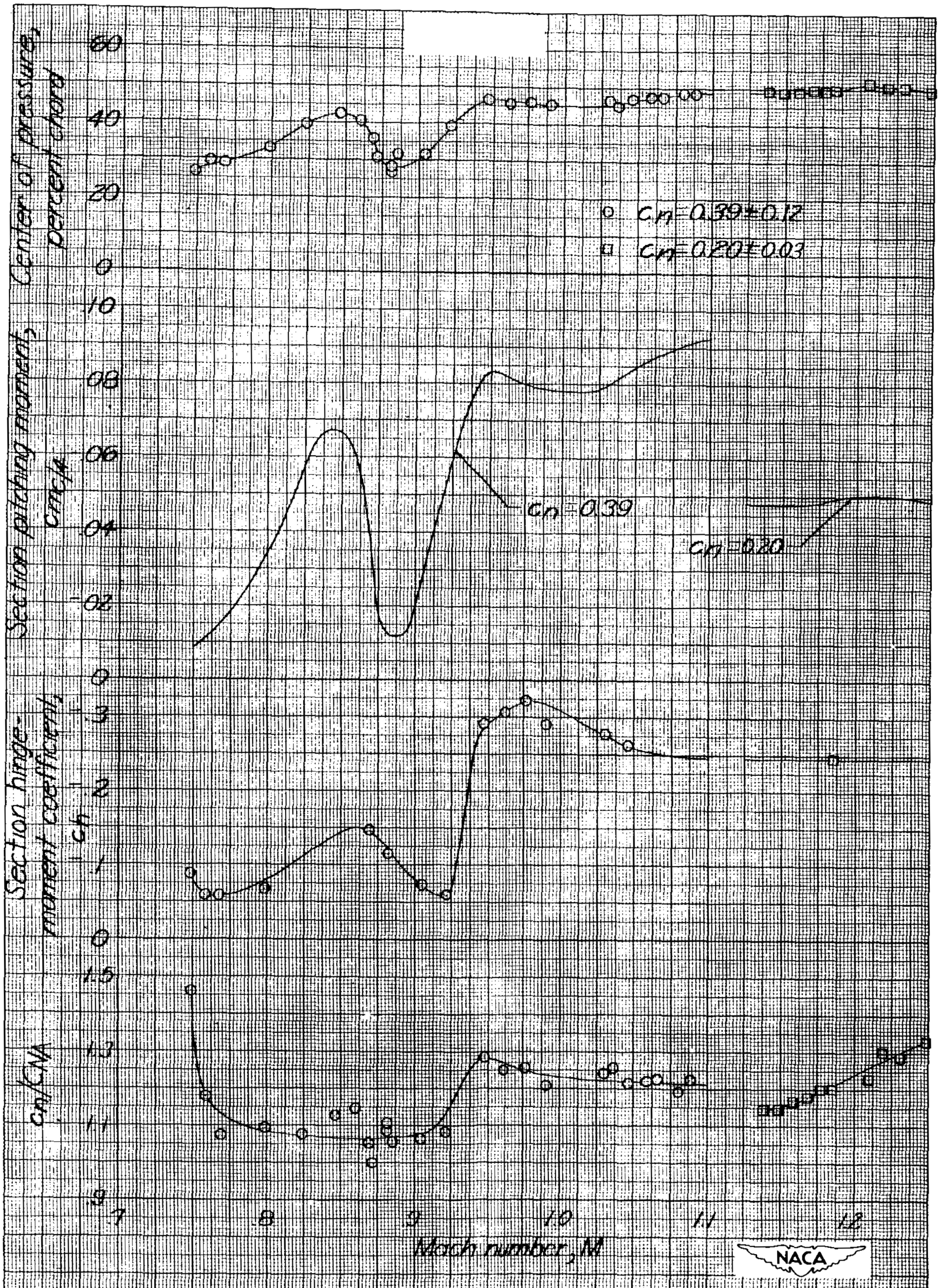


Figure 11.- Variation of the aerodynamic characteristics of the wing section with Mach number.